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DEPARTMENT OF DEFENCE

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

AERONAUTICAL RESEARCH LABORATORY

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Technical Note 32

A REVIEW OF AUSTRALIAN AND NEW ZEALAND INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1991 TO MARCH 1993

Edited by

J.M. GRANDAGE

Approved for public release.

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SUMMARY

This document was prepared for presentation to the 23rd Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Stockholm, Sweden on June 7 and 8, 1993.

A review is given of the aircraft fatigue research and associated activities which form part of the programmes of the Aeronautical Research Laboratory, Universities, the Civil Aviation Authority, the Australian aircraft industry and the Defence Scientific Establishment, New Zealand. The review summarises fatigue-related research programmes as well as fatigue investigations on specific military and civil aircraft.

DSTO AUSTRALIA

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CONTENTS

9.1 INTRODUCTION

This review of Australian and New Zealand aeronautical fatigue work in the 1991 to 1993 period comprises inputs from the organisations listed below. The author acknowledges these contributions with appreciation. Inquiries should be addresed to the name of the person identified with each section.

ARL	Aeronautical Research Laboratory, 506 Lorimer Street, Fishermen's Bend, Victoria, 3207.
CAA	Civil Aviation Authority, P.O. Box 367, Canberra, ACT, 2601.
CRC-AS	Cooperative Research Centre - Aerospace Structures, 506 Lorimer Street, Fishermen's Bend, Victoria, 3207.
MU	Monash University, Wellington Road, Clayton, Victoria, 3168.
RMIT	Royal Melbourne Institute of Technology, GPO Box 2476V, Melbourne, Victoria, 3001.
RAAF	Royal Australian Air Force
DSE	Defence Scientific Establishment, Auckland Naval Base, Auckland, New Zealand.

The previous two years have seen the formation of several Cooperative Research Centres including the CRC-AS. The main research area of the CRC-AS is the development of composite materials for application to aerospace structures.

9.2 FATIGUE PROGRAMMES ON MILITARY AIRCRAFT

9.2.1 F/A-18 Fatigue Investigation

9.2.1.1 ARL F/A-18 Full-Scale Fatigue Test (A.D. Graham - ARL)

The development of the ARL F/A-18 IFOSTP (International Follow-on Structural Testing Project), being carried out in collaboration with Canada, is progressing well, with an anticipated test start late in 1993.

The test rig has been completed, with the engine removal system and the pneumatic reticulation system the only test rig areas yet to be completed. The test rig supports an array of five air bag pairs for each horizontal stabilator, an array of ten pairs for each vertical fin, and a fuselage and engine pneumatic loading system incorporating a further twelve double acting air bag actuators. The hydraulic loading system has been completed and engine thrust loading, horizontal stabilator 'drag' loading, air brake loading and active reaction control which supports the test article, have all been successfully trialed.

The Northrop manufactured test article, FT46, has been modified for testing, with a dummy forward fuselage and test loading fixtures attached. Scrap engines have been modified to duplicate the dynamics of in-service engines, and the completed test article has been installed in the rig for initial ground vibration testing.

Over 1350 strain gauges have been installed on the test article, duplicating many of the significant gauges on MCAIR and Northrop test articles, and on CF and RAAF flight test aircraft. A comprehensive flight loads test programme supporting this test and the Canadian IFOSTP test being conducted at Canadair was successfully completed by RAAF ARDU at Edinburgh, South Australia.

The six unit electromagnetic shaker system has been completed (each shaker has a four inch stroke and is driven by a 68KW amplifier) and the application of dynamic loads during simulated manoeuvre loading has been successfully demonstrated to the desired dynamic acceleration levels on STO1, the test article on loan from the USN. Currently STO1 is in the test rig to enable development testing of the dynamic and manoeuvre loading systems to continue. An ARL developed state-of-the-art digital control system is in the later stages of completion, and single channel control on the horizontal stabilator development rig has been successful.

An extensive wind tunnel testing programme has been completed at ARL in support of the loads development. Aft fuselage and empennage aerodynamic pressure distributions were obtained for a matrix of Mach numbers, angles of attack (AOA's), side slip, horizontal stabilator angles, and trailing edge and leading edge flap settings from a full 1/9th scale model in the low speed tunnel. Data from a 1/24th scale half model in the transonic tunnel provided pressure distributions for the higher Mach numbers.

Extensive fleet usage analysis has provided 300 hour data sequences representative of RAAF and CF usage for both the pre-LEX (leading edge extension) fence and post-LEX fence periods of F/A-18 operation. Rationalized test sequences will be derived from these for each test, so that both countries can use the results of each test for their respective fleet management. The development of test loads for the ARL test centres around the use of the horizontal stabilator and vertical fin strain turning points and real time flight data contained in the usage sequences, from which inertial and aerodynamic root bending moment are derived. The wind tunnel pressure distributions and inertial mass distribution data for the F/A-18 provide the mechanism for calculating control surface load distributions.

The dynamic test loads are being derived from extensive analysis of the Canadian Aerospace Engineering Test Eastablishment data obtained during the CF aft fuselage and empennage buffet environment flight test programme. Power spectral density data have been produced for the full range of AOA's and dynamic pressures experienced during in-service flying. ARL's involvement in the NASA Ames F/A-18 full scale wind tunnel work has provided additional data for the definition of the dynamic loads at high angles of attack. An extensive coupon testing programme has been established in support of spectrum development (see below).

The investigation has generated a large number of internal reports and several official publications [1,2,3]. Fig. 1 illustrates the test rig.

9.2.1.2 Coupon Fatigue Tests (J.M. Finney - ARL)
An extensive coupon test programme is underway to support the rear-fuselage full-scale fatigue test of the F/A-18 at ARL. Initial coupon tests are helping to define the load sequence to be applied to the full-scale article in several ways, namely:

(a) producing life ratios for scaling F/A-18 test lives obtained under other sequences,
(b) giving experimental information on the influence of buffet loads on fatigue life,
(c) ensuring a fractographically readable sequence, and
(d) giving confidence in life prediction models.

Later tests will help scale the full-scale test life to particular aircraft loadings where they are different from those used in the full-scale test.

The coupon tests simulate four main rear fuselage locations using appropriate specimen geometries, materials and load sequences. The full-scale test will include both manoeuvre and buffet loads, the latter being excited by shakers. When testing coupons in a standard servo hydraulic test machine there is obviously no distinction between the origin of the loads in relation to the method of application and control. Because some of the sequences are quite lengthy (those with buffet can have a repeat length of more than half a million turning points), it is important to apply the 'buffet' cycles at a higher than normal testing frequency. Higher test frequencies can lead to larger differences between input and output amplitudes and to larger phase differences, and to overcome these problems an adaptive procedure is being used. The basis of this procedure is, firstly, to apply the required loads as a command sequence and to measure the resulting or actual loads achieved, then secondly to add the difference, for each turning point, to produce a new command sequence. This process is repeated until the errors between the required loads and the resulting loads are acceptably small.

9.2.1.3 Life Prediction Models (J.M. Finney - ARL)
Fatigue life prediction models are required for several reasons. For the coupon testing they are used to predict lives for cases that cannot be examined by test because of time constraints, and also to examine which cases might be examined experimentally. For the full-scale test they will be used in predictions for regions not covered by coupon tests, and also for sensitivity studies which may include the effect of spectrum variations or stress scale variations.

Two McDonnell Douglas numerical models are being used for these purposes [4,5]. One, CI89, predicts life to crack initiation (taken as a crack 0.25 mm long), and is based on hysteresis counting of the stress sequence and the use of constant-amplitude strain/life data as failure (crack initiation) criteria. The other model, CG90, predicts crack growth and is based on a crack face contact concept for determining the portion of any cycle effective in propagating the crack, that is, the portion of the cycle for which the crack faces are not in contact. A limited ARL appraisal of CG90 and its earlier version [6] has shown that reasonable predictions are obtained in some cases but the model appears unconservative at high negative stress ratios and for some variable-amplitude sequences. Also the effect of underloads does not appear to be well modelled.

9.2.1.4 F/A-18 FS 488 Bulkhead Fatigue Test (M. Stolz - RAAF/ARL)
A fatigue critical area near the lower wing attachment hole (the outboard mould line) of the FS488 bulkhead was identified in early fatigue testing by McDonnell Douglas. Several redesigns of this area led to different profiles and surface treatments. Northrop has carried out additional stand-alone bulkhead fatigue tests using the design usage spectrum (representing USN usage) to verify the redesigned configuration. (See Fig. 2). ARL has also

undertaken a stand-alone bulkhead test to the USN spectrum of an intermediate configuration bulkhead which was reworked during the test to represent retrofit incorporation of the production configuration change.

ARL now plans to proceed with a fatigue test on a stand-alone FS 488 bulkhead with a 6 inch radius and shot peening under a composite Canadian-Australian usage spectrum. The surface treatment and spectrum differ from those used in previous tests. The purpose of this test is to determine a characteristic life of the FS488 bulkhead with a representative configuration and tested to a representative usage spectrum prior to the Canadian full-scale fatigue test on the F/A-18 wing and centre fuselage, which is part of IFOSTP and is due to commence in December 1993.

9.2.1.5 Assessment of Cracking in F/A-18 Bulkhead Fatigue Test (G. Clark - ARL)
The assessment of fatigue cracking which occurred in the full-scale fatigue test of a stand-alone FS488 bulkhead from an F/A-18 aircraft has led to a number of significant conclusions. Firstly, the distribution of cracks at the end of the test - many small cracks distributed across a wide range of details - is testimony to the efficiency of the design and to the safe-life approach adopted. A second conclusion is that crack growth effectively commenced at the beginning of the test; it was possible to read crack growth markings back to the first applied program of loading. Thirdly, the efficient design, and the multitude of small cracks which result towards the end of the test life indicate that any possible requirement for life extension will probably not be met by moving to a safety-by-inspection approach. Finally, it was clear that the cracking behaviour seen in this test - small fatigue cracks which can occur at almost any location - carries with it an increased sensitivity to machining/manufacturing defects, in-service damage, corrosion, or metallurgical defects. A substantial program of coupon testing is currently in place to determine other methods of managing any areas of the structure which might require an increase in fatigue life; this programme addresses the effects of glass bead peening on fatigue life of aluminium alloys, and has already indicated that peening using anything other than very tightly controlled conditions may produce a reduction in fatigue life. Several publications are relevant to this work [7-9].

9.2.2 F-111 Fatigue Investigation
9.2.2.1 Boron/Epoxy Doubler Development (L. Molent - ARL)
The doubler system referred to in previous Reviews has been installed on the majority of the RAAF fleet. To date (February 1993) fifteen aircraft have been reinforced with boron/epoxy doublers. There were problems associated with three aircraft reinforced early in the programme leading to failure during cold proof load test (CPLT). A brief description of the early prototype doublers and the current production doubler system is given below.

- Flying Prototype.
  In October 1988, one set of doublers was applied to the starboard wing of aircraft A8-140. This application was conducted to verify the on-aircraft application procedure. Periodic monitoring of these doublers, during approximately three years of service, revealed no signs of deterioration. As these were considered early prototypes, the doublers were replaced by later production doublers prior to this aircraft undergoing CPLT.
- Early Production.
  Aircraft A8-148 received the first of the production doublers, followed by aircraft A8-144. Failures occurred in both doubler sets at high load levels during CPLT. Both sets of wings were considered to vary significantly (geometrically) from the specification drawings. Post-failure investigations revealed some deficiencies in the doubler application process and material system, and these were subsequently addressed.
- Current Production Doubler.
  Several modifications were made to the materials system to alleviate the high stresses in the adhesive and interlaminar regions.

The current production doublers have been applied to twelve RAAF aircraft and no CPLT failures have occurred since the above measures were implemented.

From the results of numerous strain surveys (see below) and analyses of a boron/epoxy reinforced WPF, it has been shown that substantial strain reductions were achieved. In the critical region, the relative strain reductions were approximately 40% and 20% for the positive and negative load cases respectively. The measured strain reductions were in good agreement with those predicted by finite element analysis. The reinforcement should achieve the initial design requirements, that is wing pivot fitting (WPF) survival in the CPLT, when applied to a fleet aircraft. The fatigue life of the WPF will also be improved. An overview of the doubler development programme is given in [10].

9.2.2.2 Strain Surveys (L. Molent - ARL)

Several strain surveys have been carried out in support of doubler development and for other reasons. Data have been obtained firstly from a single wing located at ARL and secondly from a complete aircraft (A8-113). The ARL wing had failed (at a repair located remote from the regions of interest) during an earlier strain survey. This failure was repaired [11] by splicing a structural steel extension to the WPF. This wing was successfully loaded to 7.3g and -3.0g [12]. The wing was then reinforced with the boron/epoxy doubler system, and a cold box was constructed around the WPF [13]. The wing was then loaded under CPLT conditions. The wing successfully completed down loading to -2.4g, but at approximately 90% of 7.3g up loading, considerable non-linearity was evident in the data from strain gauges that had been placed on the WPF. The doublers still provided some strain reduction (approximately 24%) in the critical WPF areas, when the load was raised to 98.1% of 7.3g. As load was increased further, the test rig failed slightly above 100%. The doublers remained physically attached to the WPF, but upon their removal, it was evident from the adhesive that large areas of 'poor bonding' were present (see [13]). Given the uncertainties inherent in the 'built up' wing, it was decided to investigate the procurement of a new test article. This new test article is currently being installed into the test rig.

Aircraft A8-113 was extensively strain gauged and was then strained surveyed during routine CPLT at the Sacramento Air Logistics Centre [14]. The results of the survey indicated no permanent set in the stiffener runouts, as predicted by analysis, indicating that the boron/epoxy doublers were effective in reducing the extent of yielding at these locations. All data gathered from gauges on the doublers were linear, indicating no deterioration of the adhesive bond.

9.2.2.3 Elastic-Plastic Finite Element Analysis of F-111C Wing Pivot Fitting (D. Rees - ARL)

The wing pivot fitting (WPF) of the F-111C is susceptible to fatigue damage at the number two stiffener runout and at fuel flow vent hole thirteen. In support of a Durability and Damage Tolerance Assessment (DADTA) an elastic-plastic finite element analysis of both regions has been undertaken. Since both regions are subjected to large plastic strains during the cold proof cycle, the unified constitutive model previously developed for Al7050 has been employed in the analysis of the D6AC steel of the WPF. This has been necessary because of the inability of classical techniques to adequately predict the material behaviour during cyclic plastic yielding, as is the case during the cold proof test. Initial results have compared well with data obtained from full scale wing tests conducted at ARL and should result in a more accurate estimate of the stress response for use in estimating fatigue life. Work on implementing the unified constitutive model into the finite element code is continuing and will eventually enable routine analysis of similar problems.

9.2.2.4 Other Structural Aspects of F-111 (J.M. Grandage - ARL)

The programme to estimate stiffener runout inspection intervals, which was described in the last Review, has been completed [15]. Work currently in progress at ARL will enable a reappraisal of these results.

Following failures of the horizontal tail pivot fitting during proof load testing (CPLT), the RAAF have reworked the relevant holes on all aircraft which had flown in excess of a defined period since CPLT. ARL advised the RAAF on the extent of this safe period following CPLT.

The RAAF has contracted General Dynamics (GD) and Hawker de Havilland Victoria (HdHV) to conduct a Durability and Damage Tolerance Assessment on the F-111 structure under RAAF operating conditions. As part of this programme HdHV has provided GD with loads and stress spectra for various structural locations and crack growth calculations are now in progress at GD.

9.2.3 Pilatus PC-9/A Full-Scale Fatigue Test (R.G. Parker - ARL)
As noted in a previous ICAF Review [16], the RAAF has commissioned ARL to conduct a fatigue test with the aim of establishing the service life of the fleet. The test is scheduled to commence in mid 1994. Representative loading will be applied to the wing, the empennage and the fuselage.

Although initially seen as a basic trainer, the PC-9/A is now performing much of the RAAF's advanced training role and this will be reflected in the test loading. The loading sequence will be based in part on flight strain sequences recorded from an instrumented aircraft flying representative sorties. These flight trials will occur late in 1993. The ground calibration of the flight trials aircraft is currently (March 1993) in progress. Fatigue meters are fitted fleet-wide and data from these have been analysed to investigate usage trends and to provide a basis for load spectrum development. The design of the fatigue test rig is well advanced.

9.2.4 GAF Nomad (G.S. Jost - ARL)
In the past biennium rationalisation of the overall Nomad fatigue evaluation programme has taken place. Loose ends are being brought together and emphasis is being placed on completing the more important outstanding tasks quickly.

The long-running full-scale fatigue test program has now ceased, the total accumulated flight hours being 306,000 hours. The test article has been removed from ARL to Aerospace Technologies of Australia (ASTA) for subsequent teardown inspection, the testing rig itself at ARL being now dismantled*.

Prior to removal of the test article, a static strain survey was carried out on the very comprehensively strain gauged stub-wing to aid in the design of a test specimen representing this region to be used to explore the influence of small edge-margins on fatigue life, with and without hole cold-working. Another specimen fatigue testing programme is being carried out on the wing strut to stubwing Y-fitting. On the full-scale fatigue test this fitting exhibited an unexpectedly long fatigue life: the specimen test program is intended to check the finding for this primary fatigue-critical item.

The flight testing programme referred to in the previous Review has been completed. This confirmed that the loadings applied to the full-scale fatigue test article were, if anything, slightly higher than those measured in flight: this is a conservative result so far as use of results from the full-scale fatigue test are concerned.

Major effort has been directed recently to the Nomad tailplane and its complete loading environment. This follows a fatal accident in 1990, due to tailplane failure in flight. The tailplane failed as a result of cracks which had propagated from the spar web central hole. This aircraft had been used for unusually long periods of single engine ground running. To establish the relative crack growth rates in this tailplane area for normal flight conditions and single engine ground running, programs of flight testing and of comparative resonance testing are being carried out by ASTA.

The flight testing program has shown that tailplane manoeuvre loadings within the flight envelope are less than the loads induced during single engine ground running.

ASTA are continuing with a resonance testing program which to date has shown that a pre-existing crack in the spar web central hole grows minimally under symmetric resonance loading (as in normal flight envelope operation) but grows rapidly during asymmetric resonance loading simulating single engine ground running.

A reinforcement modification for this area of the tailplane has been designed by ASTA and fitted to all civilian Nomad aircraft.

9.2.5 P-3 Refurbishment/Life Extension (J.M. Grandage - ARL)
In 1991 the RAAF developed a proposal to refurbish the avionics of the P-3 fleet and to extend its life substantially beyond the planned retirement date. ARL was tasked to consider the fatigue implications of the proposed life extension. The resulting investigation concluded that, conditional on defined operational practices and also on a structural fatigue work programme being implemented, the life extension could be justified. In part this was based on a major weight reduction arising from installing state-of-the-art avionics equipment. The investigation re-assessed the early fatigue substantiation of the P-3C and reviewed the available empirical evidence on P-3 fatigue performance in service [17].

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* For those with long memories, the main framework of this rig was that used in fatigue testing the 200-odd Mustang wings in the 1950s and early 1960s.

9.2.6 Helicopter Structural Fatigue (D.C. Lombardo - ARL)

The Aeronautical Research Laboratory (ARL) has a long history of provided advice and support to the Australian Defence Force (ADF) and, in particular, the Royal Australian Air Force (RAAF). This support has been mainly concentrated on structural integrity aspects of the fixed-wing aircraft operated by the RAAF. However the recent procurement of 39 Sikorsky S-70A-9 Black Hawk and 16 Sikorsky S-70B-2 Seahawk helicopters has raised the question of similar activity relating to helicopter structural fatigue.

Since 1991, the following tasks have been or are being undertaken in order to gain the required in-house expertise at ARL in the field of helicopter structural integrity, and to apply this expertise:

- Literature Review
  As a first step, a literature review of significant fatigue loading actions, fatigue design methodologies and structural usage monitoring was conducted and has been published as an ARL report [18].
- Attachment to U.S. Army
  An ARL engineer was attached to the US Army Vehicle Structures Directorate, (located at the NASA Langley Research Center, Virginia, USA), for a period of one year. During the attachment, the engineer worked on developing a low-cost usage monitor for the Sirkosky Black Hawk helicopter. This task provided knowledge in helicopter structural fatigue as well as helicopter usage monitoring programs [19].
- Black Hawk Fatigue Design
  A study of Sikorsky's fatigue design methods for the Australian S-70A-9 Black Hawk helicopter has been completed [20]. The study was performed to obtain an understanding of how Sikorsky undertook the fatigue qualification for the helicopter.
- Quantitative and Qualitative Usage Monitoring for the RAAF
  Some time ago, there was a firm proposal to collect quantitative data on a small sample of S-70A-9 Black Hawk aircraft under what becomes known as the Mission Severity Assessment (MSA) programme. The MSA program was meant to determine the type of flying that Black Hawks were being subjected to in Australian Regular Army (ARA) service with the aim of substantiating component lives. Although the Black Hawk is operated by the ARA, airworthiness responsibility, and hence supervision of the MSA programme, rests with the RAAF. Recently, the RAAF decided to postpone implementation of the programme. However, the RAAF will conduct a fleetwide pilot questionnaire according to specifications laid down by Sikorsky. The survey is intended to enable a revised usage spectrum to be obtained for ARA operating conditions. Further, the RAAF proposes that the revised spectrum then be used by Sikorsky to recalculate retirement lives for fatigue-critical components. While the limitations of a pilot questionnaire are well known, the cost and preparation time are considerably less than for a quantitative data collection programme. The pilot questionnaire is intended to elicit fleetwide information on the Black Hawk which will assist the definition and planning of a quantitative data collection programme. The need to proceed with the MSA or a similar programme will be reviewed after completion of the questionnaire program.
- Component life sensitivity studies
  As a complementary exercise, ARL is examining the degree of susceptability of S-70A-9 component fatigue lives to changes in the mission usage spectrum. Sikorsky's life substantiation reports, which are the basis for the currently specified retirement lives, provide the main source of input data for this examination.
- Integrated monitoring systems
  In line with other operators, the ADF is looking at the usefulness and cost-effectiveness of fitting integrated cockpit voice and flight data records, and/or maintenance data (health and usage) monitors. A working party involving membership of the ADF and ARL has been set up to look at the issues involved.

9.3 FATIGUE OF CIVIL AIRCRAFT

9.3.1 Nord 298 Propeller Blade Failure (S. Swift - CAA)

In February 1993 a Nord 298 Mohawk with 22 passengers on board suffered a propeller blade failure 10 minutes after take-off from Sydney. Fortunately the departing blade missed the fuselage. The out-of-balance forces from the damaged propeller broke two engine mounts and destroyed the engine. On the ground it was discovered that another blade from the same propeller was also cracked.

The cracks started in the bore, near the blade root, and were confirmed as having grown by fatigue. Cracks in these old style blades are difficult to detect, a fact recognised by the propeller manufacturer who had earlier recommended that operators replace old style blades instead of continuing with unreliable inspections. The blade that failed was of the old style and was being inspected, but infrequently, because the operator had argued that over a long period no cracks had been found.

This same aeroplane, which is 27 years old, had earlier suffered other major cracking including a broken stringer with a four inch skin crack in the wing. Only a rigorous fatigue management plan can keep such an old aeroplane safe.

9.3.2 Fatigue Cracks in Light Gauge Primary Structure (S. Swift - CAA)
The CAA has noticed an increasing number of important cracks in light gauge primary structure, such as the spars of control surfaces on general aviation and commuter aircraft. Such cracks not only reduce static strength; the loss of stiffness can also reduce flutter margins.

It has been disturbing that even large cracks have been difficult to find because they are often concealed by hinge brackets and doublers. Not surprisingly then, fleet-wide problems can remain hidden for years until fortuitously exposed. When the discovery of a large crack in a Bandeirante aileron spar caused an aircraft maintenance engineer to look at the rest of his airline's fleet, he found eight out of eight similarly cracked. His defect report to the CAA prompted the immediate inspection of the rest of the Australian fleet and 90% of ailerons were found to be cracked.

9.3.3 Engine Disc Failure (S. Swift - CAA)
In mid 1992, a Boeing 727 of Ansett Airlines of Australia experienced an uncontained failure of its centre Pratt and Whitney JT-8D engine during take off on a scheduled passenger flight from Brisbane. There was massive disruption of the engine, including ruptured fuel lines which torched flame for some seconds, and it was only by good fortune that the aircraft did not crash with the loss of many lives. The accident was caused by a fatigue failure of the rim of a fan hub. The crack originated in a previously unrecognised high stress field, and at a very slight scratch on the surface which occurred during an abrasive blending operation in manufacture. Numbers of other engines have been found to have similar scratches and some cracks. Two other engine failures seem to have been associated with the same type of fault. The engine manufacturer has conducted an extensive series of fatigue tests to explore the problem and airworthiness control measures are in place.

The lessons to be learned are that stress analyses are seldom exhaustive, and that a manufacturing process which was properly carried out and which would be expected to enhance fatigue life actually had the opposite effect: that it took eight years of operation to reveal the problem in this case.

9.3.4 Tail Boom Cracking in a Bell 212 (S. Swift - CAA)
In May 1992, a refueler luckily observed a 60mm crack on the side of a Bell 212 tail boom, just forward of the fin. Further investigation revealed that the crack extended circumferentially under the overlapping skin doubler up the left side, and across 90% of the top surface. Total crack length was 335mm. A longeron was also severed. A second crack had initiated on the adjacent lower skin, across the lap joint, and was propagating rapidly towards the underside. Apart from the 60mm observed externally, this crack was sandwiched between the external doubler and the internal gear box mount and longeron and was not readily inspectable (See Figs. 3 and 4).

Based on metallurgical investigation, it was estimated that the crack had been growing for approximately 2500 flight cycles, and that at the time it was found the crack was propagating 5mm every hour at each of the two crack fronts. The area had been thoroughly inspected, internally and externally, only 62 hours before, so the discovery of a crack this size in a well maintained conventional structure came as a major surprise.
A further aspect of this problem was the discovery of poor edge distances on the lower lap joint. Five holes in the underlapping lower skin actually had more hole off the skin than on. The manufacturer has conducted a finite element investigation into the effect of this, however the result suggests that the absence of edge distance was probably not contributory. A visual inspection of the rest of the Australian fleet did not reveal any other instances of similar cracking, although it needs to be remembered that a 100 - 150mm crack could remain undetected by this inspection. It is still too early to conclude that this was an isolated defect, and the investigation is continuing.

9.3.5 Dauphin Tail Boom Finger Doubler Cracking (S. Swift - CAA)
In another very fortuitous discovery, during a walk-around inspection a crack was found in the structure which joints the fenestron to the tail boom on an Aerospatiale Dauphin helicopter. Only 40mm of crack was visible from the outside, but disassembly exposed the full extent of the crack which was 395mm long - more than half way around the joint! (See Fig. 5). What was frightening was the recognition that if the crack had not emerged for 40mm, before again being hidden under the attachment plate, the crack would not have been found until the accident investigators searched through the wreckage.

An urgent X-ray inspection showed that out of the five Australian helicopters of this type, and despite several different flying roles, four were similarly cracked. It is puzzling that the French manufacturer has not heard of this problem elsewhere in the world.

9.3.6 Corrosion in Aero Commanders (S. Swift - CAA)
Recent wing spar problems in Aero Commanders should remind us not to become too blinkered when trying to control fatigue. Fatigue is not the only enemy of durability. For example corrosion can be just as destructive. Sometimes the cure for one can agravate the other.

In the 1960's fatigue in the spar caused several wing failures in Aero Commanders. The manufacturer's solution was a reinforcing strap to reduce the stresses. The production version of the strap was installed inside the wing, sandwiched between the skin and the spar, and was made from stainless steel to keep it as thin as possible. But with only minimal protection between the stainless steel strap and the aluminium alloy spar (galvanically more active), it is not surprising that corrosion of the spar is now common. (See Fig. 6). Many Aero Commanders will never realise the longer fatigue life that the reinforcing strap was designed and fatigue tested to give - they rot before they crack.

An interesting complication is the recent discovery that some wing spars were made from extrusions which have an abnormal recrystallised grain structure. Though not initially worsening the fatigue behaviour, the fine grain structure does allow stress corrosion cracks to propagate more readily. In Australia, a thirteen year old aircraft which had flown only 3300 hours was found with the aluminium lower cap of the spar cracked to the extent of 90% in both wings.

In the quest for durability, corrosion should not be ignored.

9.3.7 Janus Glider Wing Fatigue Test (L.A. Wood - RMIT)
Progress on this test has been reported in previous ICAF Reviews. The test specimen comprises a complete tip-to-tip wing assembly. The starboard wing was badly damaged in a major accident, and has been fully repaired using a variety of techniques. The port wing was purchased new from the factory.

The full scale fatigue test was stopped at 28346.6 simulated hours in October 1991 due to failure of the starboard root rib. The failure was of a catastrophic nature which appeared to be in compression bearing mode at both front and rear spigot bearings. The failure which occurred shortly after the 28000 hours inspection has caused concern due to the lack of visual development in deteriorating areas. These areas were the whitened regions below the bearing housings which have been monitored since July 1991 at 27379 hours. The root rib had been repaired previously at these locations because of crazing and cracking, which had eventually allowed the bearing housings to become loose in the rib. A summary of the root rib lives is shown below:

	Root rib fatigue lives (test hours)	
	Front bearing	Rear bearing
Original Manufacture	18743.6	12862
Repair scheme no. 1	-	1236
Repair scheme no. 2	9603	13248

An investigation into the failure has been ~~carried out by~~ the Aeronautical Reserach Laboraory, with input from the Royal Melbourne Institute of Technology, the Glider Federation of Australia and the Civil Aviation Authority.

An initial examination of the failure indicated that the bearings had separated from their housings and they remained on the spar spigots, while the housings were forced outboard through the rib into the wing box. Subsequent examination revealed the following:

1. The failure mode was fatigue, driven by shear loads transmitted through the spigot bearings.
2. Detection for delamination growth during testing was limited, possibly due to the rovings incorporated into the repair. However, deposits of finely pulverised epoxy in the wing indicated that failure had occurred over a period of time.

3. Fretting products were found at the housing and root rib interface, indicative of movement of the housing in the root rib. Wear patterns on the housing suggested that the bearing was not aligned in its housing at the time of failure.

The root rib has been repaired in accordance with the designer and manufacturer's repair scheme by the authorised repairer, Gippsland Aeronautics. Strain gauges have been installed on both the root rib and dummy fuselage, to enable the monitoring of loads transferring into these areas.

The Glider Fatigue Project at this stage is limited by funding, which is currently restricting the test to a total of 36000 hours.

9.4 FATIGUE-RELATED RESEARCH PROGRAMMES

9.4.1 Optimisation of Stresses around Holes using Sleeve/Bolt Combinations (M. Heller - ARL)

It is well known that the magnitude of alternating stresses at fastener holes in aircraft structures can be significantly reduced by using interference-fit bolts. The level of stress reduction achieved depends on the relative moduli of the plate and the insert material (sleeve or bolt) and on the local geometry. The benefits of combining an interference-fit bolt and a sleeve for a hole in a remotely loaded plate have been investigated [21]. Finite element analyses were used to study the effect of different relative moduli and different sleeve thicknesses on the induced interference stresses and the alternating stresses in a large plate. It was shown that for a given bolt material and interference level, a sleeve of higher modulus than the plate gives rise to reduced alternating stresses, Fig. 7 (shows the alternating stresses only), and increased interference stresses compared to an open hole. The inverse applies if the sleeve stiffness is less than that of the plate. An optimised balance between the interference and alternating stresses at the edge of the hole can be achieved. The implications for the more realistic case of a fastener hole close to an edge were also examined. In this case it is beneficial to use a low modulus sleeve to reduce the high induced stresses at the hole edge nearest the plate edge.

9.4.2 Fatigue Life Enhancement of Cracked Holes using Intereference Fitting (Y.C. Lam - MU)

The effect of interference fit on the fatigue performance of cracked holes has been investigated and compared with that due to cold expansion [22]. Preliminary results indicated that at low R-ratios cold expansion was more effective in retarding crack growth, but its effectiveness decreased rapidly with an increase in R. The effectiveness of interference fit in retarding crack growth was not influenced by R and consequently it is more effective at high R than cold expansion.

9.4.3 Analytical Formulae for Calculating Stresses in Unidirectional/Cross-Ply Unbalanced Laminates (M. Heller - ARL)

A multilayer theory has been developed for unidirectional/cross-ply unbalanced laminates subjected to surface shear tractions, using the assumption of plane strain [23]. The analytical solution for the stress field in the laminate was obtained and it was shown that the stresses in each layer are in good agreement with the results of a three dimensional finite element analysis. Approximate formulae for the analytical stresses were also derived for a laminate subjected to an offset distributed force at one end. The accuracy of this solution was shown by comparison with a three-dimensional finite element analysis.

9.4.4 Life Improvement of Multi-layer Joints (J.M. Finney - ARL)

Cold expansion of holes as a means of enhancing fatigue life in aluminum alloy specimens becomes progressively less effective as the specimen becomes more complicated. In multi-layer joints the life improvement by cold expanding holes through the whole thickness appears to be about one-third that in single-layer, open-hole specimens. In the multi-layer case it appears that fretting initiation of fatigue cracks may prevent the full benefit of cold expansion being realised. The use of fasteners with an interference fit might prevent fretting and hence improve life. The results of experimental work aimed at examining the influence of fastener interference on the fatigue life of multi-layer aluminium alloy specimens with cold expanded holes, are given below; the experiments used hole expansions of zero and 4%, and fastener interferences of zero, 0.5% and 1.5%. All tests were made with an F/A-18 bulkhead flight-by-flight load spectrum.

For low-load-transfer specimens the following broad conclusions apply:

(a) With neat-fit fasteners, cold expansion improves spectrum fatigue life by a factor of about four.
(b) In the absence of cold expansion, interference-fit fasteners give a life improvement factor of about six.
(c) There is no effect of cold expansion in the presence of interference-fit fasteners.
(d) Interfacial fretting dominates the failure mode and this feature is clearly incompatible with good load transfer in low-load-transfer joints.

For 100% load transfer specimens where it was thought that interfacial fretting would not dominate and that the full potential for improvements in fatigue life might be obtained, the following conclusions apply (though the testing is not quite complete).

(a) With neat-fit fasteners, there is only marginal improvement in spectrum fatigue life by cold expansion.
(b) In the absence of cold expansion, progressive improvements in spectrum fatigue life are obtained with increasing fastener interference, but the maximum improvement is much smaller than that obtained with low-load-transfer joints.
(c) There is a significant effect of cold-expansion on fatigue life when combined with the use of interference-fit fasteners.
(d) Some interfacial fretting occurs in the joint but it does not appear to dominate fatigue life.

9.4.5 Fatigue Behaviour of Mechanical Joints in Composite Laminates (S.C. Galea - ARL)
Work on mechanical joints in thick composite laminates has continued. Results are contained in [24]. This work studied and mapped damage around the fastener holes in thick composite-to-metal mechanically fastened joints under MCAIR sequence loading. An ARL developed in-situ C-scanning device was used to monitor the delamination growth around the holes during the test. Other non-destructive methods used to monitor the health of the joint were load/deflection plots for joint compliance measurements and shadow Moire fringe techniques to measure fastener tilt. This work reported the investigation of wear at the fastener hole, development of damage around the hole and delamination growth. Also, bearing failure studies in thick composite laminates have indicated that a bearing type of failure occurs in regions around the fastener hole of the composite laminate close to the composite/metal fixture interface.

Preliminary studies on the effects of incipient single and multiple delamination damage around fastener holes were also studied. Both cases showed that, under similar loading conditions, a reduction in fastener fatigue lives had occurred compared to those obtained for the initial non-damaged case. A significant reduction in the fastener fatigue life was observed for the multiple delamination case.

Testing under hot/wet environments (viz., 82°C/60% RH and 86°C/96% RH) of composite-to-metal mechnically fastened joints has also been undertaken [25]. Results to date show a marked decrease in the fatigue life of composite-to-metal mechanically fastened joints, with fastener failure occurring at fatigue lives considerably less than those observed at ambient conditions under the same load conditions. The process of delamination initiation and growth appeared to be enhanced by the hot/wet conditions.

Future work will concentrate on using NDE techniques such as SPATE, thermography, piezo film sensors, shadow Moire fringe and ultrasonic C-scanning to monitor the health of the joint. The investigation into the effect of incipient damage around fastener holes on the static and fatigue performance of mechanical joints will be continued.

Studies of mechanical fastener joints using thin composite laminates has continued with the initial evaluation of a back-to-back testing arrangement. However load transfer problems caused by the asymmetrical nature of the arrangement have brought about a re-think of the specimen configuration. The new test specimen under consideration is a simple two-fastener lap-joint specimen.

ARL is also a participant in the CRC-AS along with Hawker de Havilland, RMIT, Monash University, Sydney University and the University of New South Wales. ARL, CRC-AS, Sydney University and the University of New South Wales are involved in a task on Bolted/Bonded Joints (incorporated in the Thin Skinned Structures Program). The task has three main objectives, viz.,

1. To develop a validation procedure for the design of bolted/bonded joints for thin skin constructions.
2. To investigate alternative design/manufacture of bolted/bonded joints.
3. To develop test procedures and methods to determine design allowables of joints.

9.4.6 Durability of Postbuckling Fibre Composite Panels Under Shear (M.L. Scott - CRC-AS)
An investigation employing theoretical and experimental techniques is being undertaken to assess the performance of thin composite panels in the postbuckled state. The first stage of the programme involved static shear buckling tests of 1 mm thick unidirectional and quasi-isotropic panels fabricated from T300/914 carbon/epoxy unidirectional pre-impregnated tape. A picture frame test rig was employed to load 250 mm square panels, with clamped edge conditions, in a servo-hydraulic universal testing machine. Arrays of electrical resistance strain gauges were used for surface strain measurement. Experimental buckling loads were identified and compared with theoretical buckling loads calculated using a simple graphical method and the finite element

package MSC/NASTRAN. The geometrically non-linear capabilities of NASTRAN were utilised to predict the postbuckling behaviour of the panels.

In the second stage of the programme, quasi-isotropic panels are being fatigue tested in compression-compression load control mode at a cycling frequency of 5Hz. Several nominal buckling ratios are being employed. A number of tests will also be repeated at 70°C and 95% humidity using panels conditioned to 0.7% moisture content. The results available from early tests suggest that shear buckling ratios in excess of 5 can be employed safely in thin panels designed to operate in the postbuckled state.

9.4.7 Damage Tolerance of Bonded Repairs (D. Rees - ARL)

A test programme to investigate the effects of damage in a boron/epoxy patch bonded to a rivetted lap joint specimen was conducted. The test specimen was designed to represent a fuselage lap joint typical of modern commercial jet transport aircraft. Previous testing of the specimens had investigated the phenomenon of multi-site damage growth, included measurement of crack growth rates and an evaluation of the failure mechanism. A proposed boron/epoxy repair was evaluated and shown to be effective in arresting crack growth [26,27]. Further testing demonstrated satisfactory performance of the repair under adverse environmental conditions [28]. Damage in the repair included adhesive disbonds and low velocity impact damage. Adhesive disbonds were simulated in some specimens using teflon film inserts. A constant amplitude fatigue loading was applied. Disbond damage was monitored using the Moire fringe technique and crack growth monitored using eddy current inspection. No failures occurred in the repairs and no significant damage growth was observed. Four specimens were loaded in tension to failure. The static strength of the damaged repairs exceeded that for an undamaged, unrepaired lap joint specimen. The test programme is reported in [29].

A further experimental programme has been undertaken to investigate the damage tolerance of boron/epoxy repairs to aluminium plate specimens containing edge and centre cracks. Following crack initiation and fatigue precracking, the specimens were repaired with a twelve-ply unidirectional boron/epoxy patch. Small teflon film inserts were placed between the patch and adhesive to simulate adhesive disbonds in a number of specimens. The specimens were environmentally conditioned for several weeks in a 5% NaCl solution at 60 deg C. All testing was subsequently done with the patches saturated at approximately 1% moisture content and a temperature of 60 deg C. A constant amplitude tensile fatigue loading was applied which would normally produce a fatigue life for the unrepaired specimen of approximately 10,000 cycles. After 50,000 cycles had been applied the specimens were removed for inspection. Eddy current inspection revealed that no significant crack growth had occurred since the application of the patch repair. The specimens were impacted over the crack tip and/or at the centre of the patch with an absorbed impact energy of approximately 8 Joules. The specimens will be subjected to further environmental conditioning and the fatigue loading repeated to assess the effect of the impact damage on the integrity of the composite patch.

9.4.8 Multiaxial Elastic/Plastic Response of Al7050 (D. Rees - ARL)

A constitutive model has been developed for 7050 - T7451 aluminium alloy based on the unified model of Stouffer and Bodner [30,31]. This approach is more accurate than the classical approaches which fail to adequately predict the response of modern engineering materials to multi-axial, cyclic plasticity. It has been shown [32] that the model accurately predicts the response for uniaxial monotonic and cyclic loadings and stress relaxation during periods of constant plastic strain. A series of further tests were carried out in which a number of complex uniaxial loadings were applied under strain control. These consisted of blocks of constant amplitude strain cycles separated by periods of constant strain, and also a load spectrum resulting from the sum of two low frequency sinusoidal inputs. These loadings may be considered representative of the loadings experienced by modern military airframes. The predicted response agreed well with the experimental results [33], see Figure 8. A test specimen has been designed to investigate biaxial elastic/plastic behaviour. It is designed to contain a uniform biaxial stress field when subjected to biaxial loadings, see Figure 9. Testing is to be carried out in a 250 kN biaxial test machine which will be run under strain control. The constitutive model has been modified to account for biaxial loadings and the predicted response from the model will be compared with the experimentally determined behaviour.

9.4.9 Bonded Composite Repair Technology for Metallic Aircraft Components (A.A. Baker - ARL)

The use of high strength, high modulus fibres to repair or reinforce defective aircraft structure has been shown to be a highly efficient and cost effective repair technique. Some recent work at ARL has been aimed at demonstrating the long term durability of such repairs. ARL has collaborated with Deutsche AIRBUS, Helitech Industries (the holder of the technology licence) and the FAA to apply demonstration repairs to the lap joint of the AIRBUS A340 test fuselage in Munich, Germany. The lap joint was prepared by inserting two long saw cuts in the upper and lower row of rivets. Patches were applied over these cuts which represent the multi-site type of damage commonly seen in the lap joints of fuselage structure. These patches will be monitored during the course of the fuselage test program to check for crack growth.

In another recent example, two patches were applied to a McDonnell Douglas MD-82 aircraft which had sustained fatigue cracking in the rear skin of the leading edge slat. Careful design was required in this case as the area is heated by bleed air from the engine for de-icing purposes and the expected temperatures were higher than those normally encountered on a civilian aircraft. This repair was easily applied during a routine service and did not require the removal of the slat.

9.4.10 Graphite/Epoxy Environmental Exposure Programme (R.J. Chester - ARL)
A tropical exposure trial is being carried out in association with the tropical exposure site at Materials Research Laboratory (Queensland) to determine the long term durability of graphite/epoxy specimens representative of aircraft components. The specimens are in the form of honeycomb sandwich beam specimens and are loaded outdoors in constant amplitude four point bending while being exposed to the full effects of the tropical environment. Most of the specimens have impact damage while a number have representative repairs to the graphite skin. The specimens are loaded in blocks of 100,000 cycles at a frequency of 0.013 Hz. Between these loading blocks the beams are removed from the loading rigs and subjected to thermal cycling (-50^{o}C to +150^{o}C) together with 40 over-load cycles to 3500 microstrain. These conditions are designed to simulate as closely as possible the conditions of actual aircraft usage.

Damage assessment is carried out by ultrasonic C-scanning, compliance measurements and residual strength measurements at the end of the trial. The beams have so far experienced over 800,000 cycles at 2500 microstrain and 250,000 cylces at 2750 microstrain without any evidence of delamination growth. The strain level in these tests will soon be raised to 3100 microstrain.

9.4.11 Piezo-electric Sensors for Damage Assessment of Cracks (W.K. Chiu - ARL)
Damage detection in metallic and composite structural components is of particular importance in the damage tolerance assessment and maintenance of aircraft structures. A preliminary study on the use of piezoelectric film for such assessment is reported in [34]. In this report, the ability of piezoelectric sensors in monitoring damage in three types of specimens was evaluated. They are: (1) a riveted lap joint, (2) a surface crack in an aluminium alloy, (3) an impact damaged composite laminate. This qualitative study showed that it is possible to detect various types of damage in both metallic and composite specimens using piezoelectric film.

A numerical investigation into the use of such sensors to detect and monitor fatigue crack growth in a patch-repaired structure has also been performed. This numerical evaluation showed the possibility of using piezo-sensors for the detection of crack growth under a patched structure (see [35]). The sensitivity of such a sensor for crack detection is shown in Fig. 10.

A centre-crack specimen was used to validate the constitutive relationship between surface strain and the output from a piezoelectric strip. A combined analytical and experimental analysis was performed for this purpose. An example of the theoretical results obtained is shown in Fig. 11. In addition to experiments performed using test specimens, piezoelectric sensors have also been applied in real structures. An experimental program to study the changes in load path on a full-scale P3C fatigue specimen is underway.

9.4.12 Improved Methods for Stress-Decomposition (T. Tran-Cong - ARL)
Local stress states inside a structure determine its behaviour in fracture and fatigue. A programme has been initiated to establish a method of determining the local stress states from observation of the bulk stress on external surfaces using thermal emission techniques. The feasibility of this method of determining local stress states from external observation is demonstrated in a simplified model of a rectangular prism suffering a transverse crack which creates a concentration of stresses. An analytic method can determine various types of direct stress in the plane of the transverse crack in the rectangular prism. The results are reported in [36, 37]. The method could be easily extended to other equally important cases such as a transverse crack in a circular cylinder or shear loading applied through the transverse crack plane of the rectangular prism. The solutions developed here belong to the group used to attack the still unsolved problems of prisms and cylinders under loads.

9.4.13 Simulation of Multisite Cracking (T. Tran-Cong - ARL)
Multisite cracking approximates the behaviour of many actual components. The fundamentals of this phenomenon are under investigation and computer simulation is being carried out to study the interaction of different geometrical parameters and of stress concentration factors. The aim is to establish a model which can account for the existence of many cracks with interaction.

9.4.14 Stable Tearing in Near Plane Strain Conditions (G. Clark - ARL)

Over many years, ARL staff have observed significant areas of stable tearing (i.e. microvoid coalescence fracture modes) which occur on fatigue fracture surfaces of high-strength steels and aluminium alloys from aircraft parts and in laboratory specimens. These bands are common, even where the constraint condition has been close to plane strain, leading initially to concern that the tearing fracture mode indicated incipient failure, particularly where such tear bands are visible at an early stage of crack development. However, consideration of the situations in which tearing has been observed suggests that *stable* tearing is quite common even in situations where much of the sample is experiencing plane strain conditions.

The tearing can be very extensive -- in some examples, stable tearing covers much of the fracture surface, making it particularly difficult to assess fatigue crack growth rates by surface examination, and distorting the crack growth rate results. This has been experienced recently in Australia in assessing the growth rates of fatigue cracks in a crashed Macchi aircraft; the assessment of precisely which aircraft loads caused which tear bands required substantial analysis. Crack shapes after a stable tear may be very distorted compared to those normally expected in fatigue, with large regions of "tunnelling" i.e. crack advance which is stabilised by near-surface ligaments. These observations demonstrate that a single load cycle in fatigue is capable of causing a very large increment in crack length, an observation with significant implications with respect to NDI, the development of fracture control approaches based on fatigue crack growth, and fracture criticality. The observations made at ARL support the view that the crack growth resistance curve can still be rising enough to permit substantial amounts of stable tearing (effectively a pop-in phenomenon), often repeated many times, even when constraint conditions approach plane strain. The implications of the presence of stable tearing need to be understood to ensure a correct view of how close the crack is to instability. Details are given in [38].

9.4.15 Benchmark Bending Test of a Thick Specimen (J.M. Finney - ARL)

Fatigue failure only occurs in metals with the action of reversed plasticity, usually in localized regions, and given that technology is heading in the direction of predicting cyclic plasticity by numerical techniques it becomes important to be able to assess the accuracy of numerical procedures. Cyclic plastic strains in commercial metals can be of the order of 10% or greater and it would seem reasonable to require of any numerical procedure that it should accurately predict behaviour under similar monotonic strains. To this end a bending test on a 30 mm square section 7050-T7451 aluminium specimen has been made to two levels of bending moment in four-point loading. Before deformation a square grid pattern of 2 mm spacing was applied to each of the four faces of the test section, and it is the deformations of the grid pattern that were used to define the surface displacements under bending.

The tensile stress/strain curve for the material was obtained for strains up to 0.12, and likewise, stress relaxation measurements were made at strains of 0.04, 0.08 and 0.12. These data, combined with the details of the bending experiment, are sufficient for displacement calculations. Fig. 12 illustrates one set of displacement measurements - the case shown is that of the out-of-plane displacements of the compression surface.

A report is available on the work [39], giving both experimental details and measured displacements, allowing a comparison of the results with those from numeric analyses.

9.4.16 Thermoelastic Stress Measurement (S.A. Dunn - ARL)

The previous two Reviews have detailed research involved in using the mean stress dependence of the thermoelastic parameter (K) to measure residual stresses. This mean stress dependence of K is due to the stress dependence of the coefficient of linear thermal expansion (α) which is easily shown to vary linearly with stress in the linear elastic regime. Two studies have been found ([40] and [41]) which show that the theory describing the stress dependence of α does not apply outside the linear elastic regime. In [40], the apparent discontinuous change in α as the material exceeds the elastic limit was shown for mild steel (0.1% C) and pure magnesium. This work was extended in [41] where this effect was shown for 1020 steel and an Invar. These results have great bearing on any attempts to use the stress dependence of α to measure residual stresses. This was found to be the case in [42] where an attempt to measure the residual stresses around a cold-worked hole in a 2024 aluminium plate using the stress dependence of α was made. The results presented in [42] show anomalous results in the plastic region around the hole consistent with the results presented in [40] and [41]. Whilst these results show that the measurement of residual stresses using the stress dependence of α *is* a much more difficult proposition than previously thought, the plastic strain dependence of α opens the possibility of using this to examine components for damage and overloads by investigating anomalous behaviour in thermal expansion.

The frequency dependence of the observed thermoelastic temperature changes due to thermal conduction for composite materials was also discussed in the previous Review. This thermal conduction effect for composite materials was first shown in [43] and was numerically modelled in [44]. In [44] it was also shown how the variation in the observed surface temperature changes with frequency of loading could lead to a means of the determination of strain components for a composite laminate from thermoelastic temperature data. The thermal conduction was modelled analytically in [45] and this model was successfully used to determine strain components for a composite laminate with a hole in [46].

9.4.17 Gas Turbine Engine Materials (B.J. Wicks - ARL)

Fatigue crack growth rates have been measured for (i) a conventional disc alloy, Waspaloy at 500, 600 and 700°C, and (ii) direct aged DA INCO 718 at 500 and 700°C, in air, at stress intensity factor ranges (ΔK) from 10 to 50 $MPa\sqrt{m}$ with triangular wave forms with cyclic frequencies of 2 Hz, 1/10 Hz, and 1/60 Hz, and trapezoidal wave forms with hold times at peak load of 2, 10 and 60 seconds. The effects of these variables on crack growth rates were similar for both alloys, despite differences in microstructure, e.g. a much smaller grain size for DA 718. The main trends are shown in Fig. 13 for Waspaloy. Differences in cycle frequency and wave forms had relatively small effects on crack growth rates at 500°C, but there were large effects at 700°C which depended on the ΔK value.

At low ΔK, long hold times (60 seconds) resulted in much lower growth rates (and lower threshold ΔK values) than for continuous cycling at 2 Hz or 1/60 Hz. *Lower cyclic frequencies resulted in higher crack growth rates*, with more marked effects at higher ΔK values. At high ΔK values, there was little or no effect of hold time on crack growth rates compared with continuous cycling at 2 Hz.

Extensive studies of fracture surfaces and metallographic sections through partially cracked specimens indicated that the effects of cyclic frequency and wave form could be explained in terms of a complex interplay between factors which promote crack growth and those that inhibit crack growth. These effects are summarised in the Table below.

		Inhibited crack growth due to		Enhanced crack growth due to		
		Oxide-Induced Crack Closure or Blunting	Fracture-Surface Roughness Induced Closure	Crack Branching	Creep	Slip Localisation
High ΔK	2 Hz	-	-	-	-	-
	1/60 Hz	-	-	xx	xxxx	xxx
	60 s hold	-	-	xx	xxx	-
Low ΔK	2 Hz	x	x	-	-	-
	1/60 Hz	x	xx	xx	xx	x
	60 s hold	xxxxx	xxx	xxx	xx	-

The Table summarises the relative importance of various effects on fatigue crack growth rates for Waspaloy at 700°C; increasing number of crosses indicates increasing importance. (The absence of a cross does not imply that the effect is completely absent). A similar, only slightly modified Table would be applicable to DA 718.

The implication of the results for fatigue life prediction under realistic spectrum loading is that present methods of life prediction involving separate summation of fatigue components provide relatively accurate modelling of high temperature fatigue crack growth using data derived from continuously cycling tests. For long cycle times the prediction is expected to be less accurate, but still conservative. It appears that because of a fortuitous combination of factors the fine grain size of DA718 does not lead to enhanced creep/fatigue crack growth.

Deformation behaviour and microstructural stability of the superalloy DA718 have been investigated under simulated gas turbine engine operating conditions for a turbine disc, with (i) temperatures of 650 and 700°C and at room temperature, (ii) cyclic frequencies of between 0.05 to 0.66 Hz and (iii) controlled plastic strain ranges ($\Delta\varepsilon_p$) of 0.2-0.4%, leading to failure in between 750 and 1500 cycles. Cyclic deformation at elevated temperatures results in the formation of deformation twin bands on $(111)_\gamma$. No evidence of dislocation deformation bands could be detected. The character of the microstructure appears to be relatively insensitive to changes in the frequency of testing and the plastic strain range. Some grain growth appears to occur, although the extent of this is difficult to assess owing to variations in grain size in as-received alloy. Some coarsening of γ'' particles occurs during LCF testing when the duration of the test is more than 10 hours.

This study suggests that twinning is a dominant mode of deformation during cyclic loading and that the γ'' particles are sheared by the propagation of these deformation twins. An important feature of the microstructure of LCF tested specimens is the occurrence of small platelets of δ at the grain boundaries. These platelets appear to share an orientation relationship with the matrix, and they probably form as a result of shear induced transformation of γ'' particles. The passage of dislocations through the γ'' particles might have resulted in small domains of δ which subsequently grow. Diffusion along grain boundaries is anticipated to promote the growth of the δ phase.

Gradual cyclic softening is observed during cyclic loading at 650 and 700°C. This is attributed to the coarsening of γ'' particles during testing and their transformation to δ phase. Cracks initiate commonly at blocky carbide particles located at or close the specimen surface.

Details of the above investigations are in [47 to 53].

9.4.18 Improved NDE of Discrete and Distributed Damage (C.M. Scala - ARL)

This research is aimed at developing improved techniques, based on ultrasonic and electromagnetic nondestructive evaluation (NDE), for the detection and assessment of discrete and distributed damage in RAAF aircraft components.

- Ultrasonics

The extension of aircraft operation beyond original design life requires development of improved methods of damage detection in the aircraft components. Ultrasonic nondestructive evaluation is routinely used for inspection of aircraft components. However, the usefulness of ultrasonics is limited by various features of the piezoelectric transducers conventionally used for inspection, viz. their large physical size, the considerable variability in coupling transducers to a specimen, complex directivity pattern and complicated frequency response. A laser-based system has been developed to overcome many of these problems. The laser-based system uses a Nd:YAG pulsed laser for generation, optics for beam-shaping and an interferometer for wide-band detection of ultrasound [54].

The main advantages found in using the Nd:YAG laser for the generation of ultrasound are that the ultrasonic source is highly reproducible, non-contact and wideband with easily variable properties (amplitude, size and shape, frequency and directivity pattern) which are readily quantifiable. Such a laser source has further advantages of access into tight, curved locations or in corners and has the potential to be more readily and rapidly scanned, using optics, than conventional transducers. The interferometer for detection also requires minimal access and has the potential for rapid scanning. Some of the laboratory applications where there have been benefits in using the laser-based ultrasonic system are discussed below.

Improved crack sizing

The laser-ultrasonic capability for varying source directivity pattern by adding suitable lenses was useful in sizing surface fatigue cracks about 3 mm in depth in corners. Current work is aimed at extending the work to the sizing of smaller cracks.

Corrosion detection

The ability to generate different wave types with the laser was useful in developing a technique to assess the extent of hidden corrosion damage occurring in thin painted aluminium sheet comprising part of a P3C elevator flap [54].

Transducer calibration

Use of the interferometer has also provided an accurate method for determining the initial performance and long-term deterioration of conventional piezoelectric transducers, which should lead in turn to improved choice of conventional sensors for inspection problems. The interferometer has been used to measure the displacement across the front face of a 10 MHz ultrasonic probe, as part of a TTCP round-robin calibration [54].

Composite repair/reinforcement problems
Unidirectional boron-epoxy composites are used for reinforcement of highly stressed regions in aircraft, but there is currently no NDE technique for monitoring changes in adhesive bond strength between composite overlay and the aircraft substrate. The potential use of ultrasonic leaky interface waves for the adhesive bond assessment is being examined. For the case of the overlay well-bonded to aluminium, titanium or steel substrates, leaky interface waves have been predicted theoretically for wave propagation perpendicular to the fibres [55-57]. The next stage in the work is an experimental verification of the existence of these interface waves.

- Eddy Currents

There has been significant progress in the development of theoretical models for eddy-current NDE.
Prediction of coil response
It is now possible to predict eddy current coil response in a number of idealized geometries, including first- and second-layer through-cracks, and surface-breaking cracks in plates [58]. The predictions have been verified in a series of carefully controlled benchmark experiments devised as part of collaborative research under TTCP PTP5 [59]. The benchmark experiments also provide a common set of coil and defect parameters which can be used to compare the effectiveness of different numerical models in the future.
Crack size determination
Eddy current testing is a reliable method for crack detection but is rarely used to determine crack depth due to complications in the use of calibration flaws and crack shape variability. An alternative approach considers the problem more fundamentally by developing an algorithm for crack-size determination in a plate with an infinite crack of uniform depth and uniform crack-opening. This algorithm has been tested using a set of artificial cracks. The crack depths deduced by applying the algorithm agreed with actual crack depths to within 15% or better, suggesting that this approach could form the basis for future crack size determination. Present research is now being directed towards eddy-current detection and possible sizing of cracks in fastener holes.

9.4.19 NDE of Corrosion (G. Clark - ARL)
A review of materials-related maintenance problems in the Australian Defence Force highlighted the need for improved management of corrosion in aircraft structure. Currently, such corrosion is generally managed on an on-condition basis, and there is a need for the introduction of improved preventive measures at many stages of planned maintenance, as well as the use of opportunity-based preventive treatments. A crucial problem is the difficulty of detecting and assessing corrosion in multi-layer assemblies, particularly around fastener holes; most NDE methods do not perform well in these situations, and those with some ability to handle multi-layer systems (often eddy current methods) are often confused by surrounding mechanical detail. Novel methods of NDE are currently being explored as a possible means of improving the detection capability for hidden corrosion; of main interest are Compton backscattering X ray methods and thermographic methods. These are being examined as a means of detecting corrosion in P3-C Wing Spar caps. Details are given in [60].

9.5 REVIEW OF STRUCTURAL FATIGUE RESEARCH AND DEVELOPMENT IN NEW ZEALAND

9.5.1 Andover C Mk1 (W.L. Price - DSE)
Royal New Zealand Air Force (RNZAF) requirements for detailed information on fatigue damage effects in its Andover medium transport aircraft are being met by a loads/environment spectrum survey (L/ESS). The data acquisition phase of the project, which is close to completion, has provided detailed information on airframe loads in routine operational service. Strain and parametric data provided by a microprocessor recorder have been analysed to provide measurements of relative fatigue damage rates associated with individual missions and flight sequences (Fig. 14). Preliminary indications are that the majority of flying is conducted in a relatively benign environment although some missions, particularly those involving extensive flight segments at low altitudes, can be associated with high rates of fatigue damage.

The Andover L/ESS has provided information and experience which is expected to be highly relevant to the Orion structural loads monitoring project outlined below.

9.5.2 P-3K Orion (W.L. Price - DSE)
Preparations for an L/ESS project involving an RNZAF Orion have involved a detailed review of relevant engineering data. Comparative analysis based on counting accelerometer data has confirmed that in the severe New Zealand operational environment, the P-3K fleet has consumed airframe fatigue life faster than similar aircraft elsewhere. P-3K airframe maintenance activities have included safety by inspection methods for some time but analysis had indicated that the inspection requirements may need to be reviewed and updated. The L/ESS is required to clarify the manoeuvre and gust loads spectra so that potential crack growth rates in critical parts of the airframe can be defined accurately.

Since the aircraft are likely to have to remain in service for another 20 years the RNZAF is investigating additional ways of maintaining the operational availability of the fleet over that period. Options such as the replacement of wing and empennage components are currently being assessed. The L/ESS programme will proceed when these engineering decisions have clarified the structural integrity issues which will be critical to the future operation of the P-3K fleet.

9.5.3 Aermacchi MB339C/B (S. Ferguson - DSE)
The MB339 fleet has become fully operational in the RNZAF jet trainer role. Individual aircraft tracking (IAT) is based on the use of counting accelerometer equipment which is supplemented by Aermacchi Aircraft Strain Counter (ASC) systems fitted to all aircraft. Fatigue damage rates in some airframe components have been found to be relatively high and a detailed analysis of damage rates associated with individual mission types is being conducted to assess the fatigue severity of particular types of flying. The RNZAF plans to repeat the flight by flight analysis at regular intervals during the life of the fleet to assist structural life management activities and to ensure that fatigue life consumption rates remain within required limits.

9.5.4 Fatigue Analysis (P.J. Riddell - DSE)
The increased use of operational loads data recording equipment in New Zealand has led to the development of an analysis method which permits fatigue damage to be calculated in real time [61]. The procedure has been used to assess the relative fatigue severities of individual manoeuvres and flight sequences.

Parallel research work has been aimed at reducing the empiricism of existing fatigue crack growth prediction procedures. Single increments of crack growth associated with individual load cycles have been measured (Fig. 15) during fatigue tests on specimens fitted with high resolution crack length measurement equipment [62]. Information provided by the system prediction models based on crack closure criteria need to be refined. Results obtained using a Dugdale strip yield analytical model have been promising but further improvements are needed.

Computer database software has been developed to permit crack growth calculations to use information from a large number of fatigue tests conducted under a range of different loading conditions.

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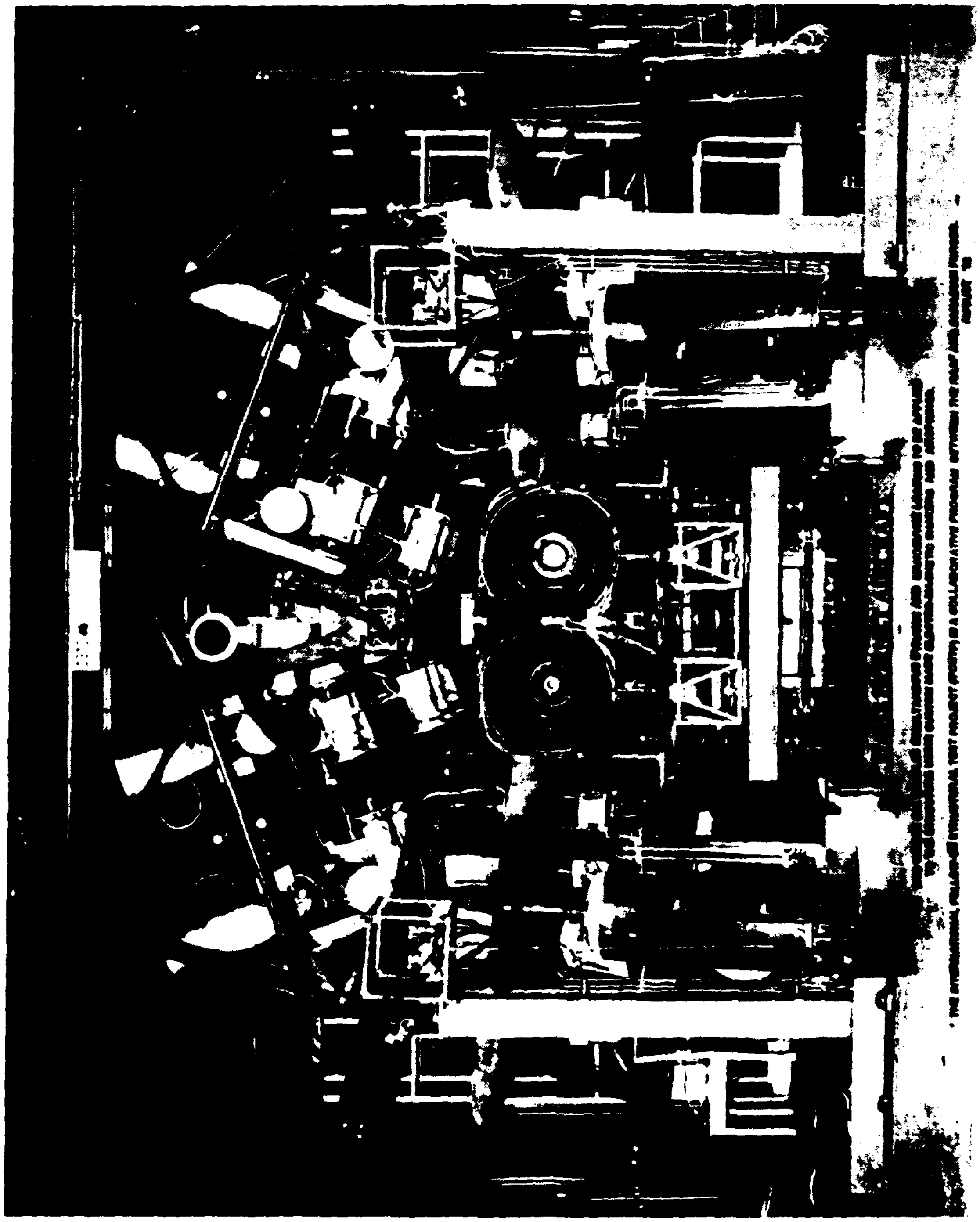


FIG.1 F/A-18 FATIGUE TEST RIG

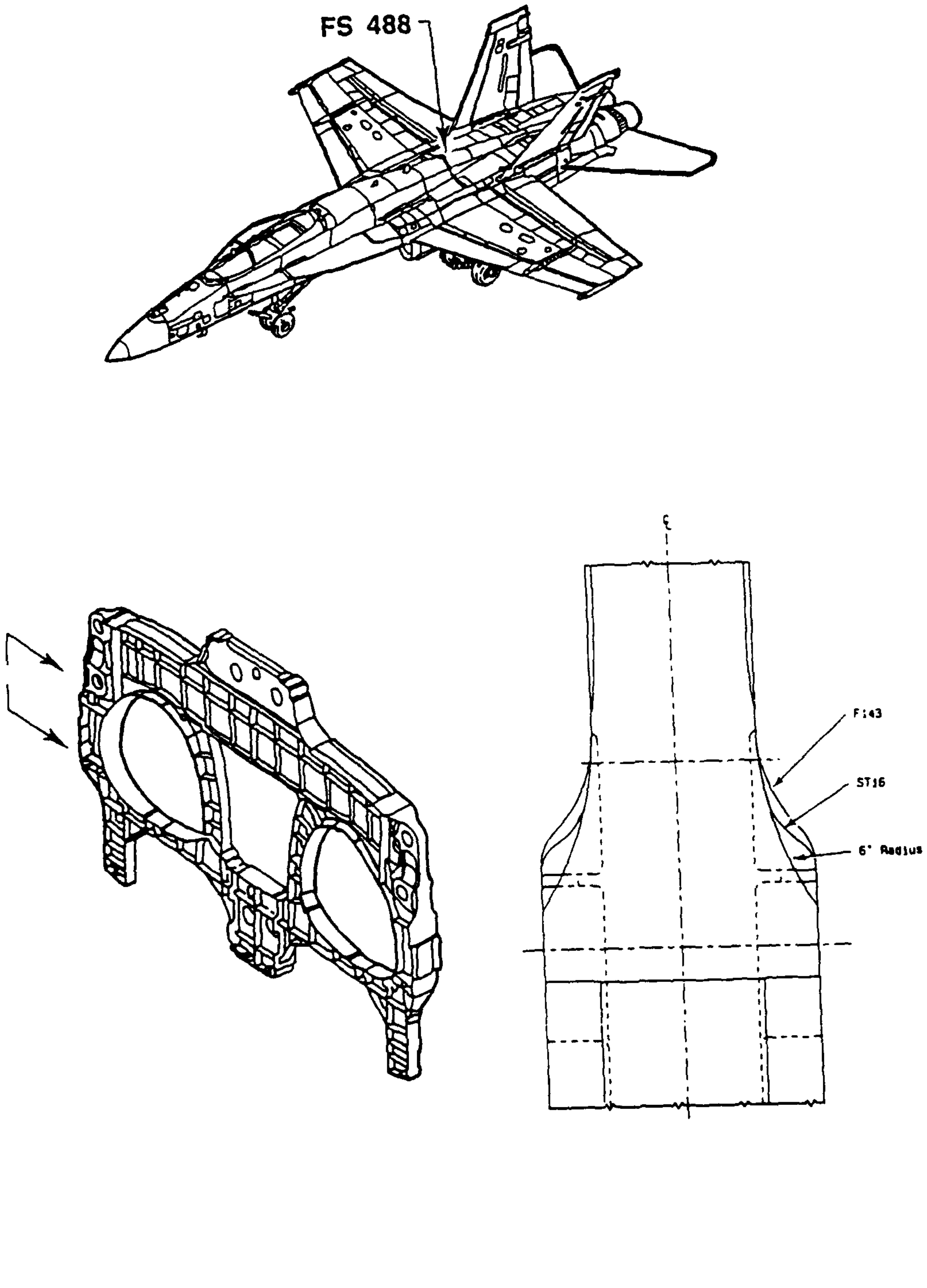


FIG. 2 FUSELAGE STATION 488 BULKHEAD, SHOWING VARIOUS FLANGE PROFILES

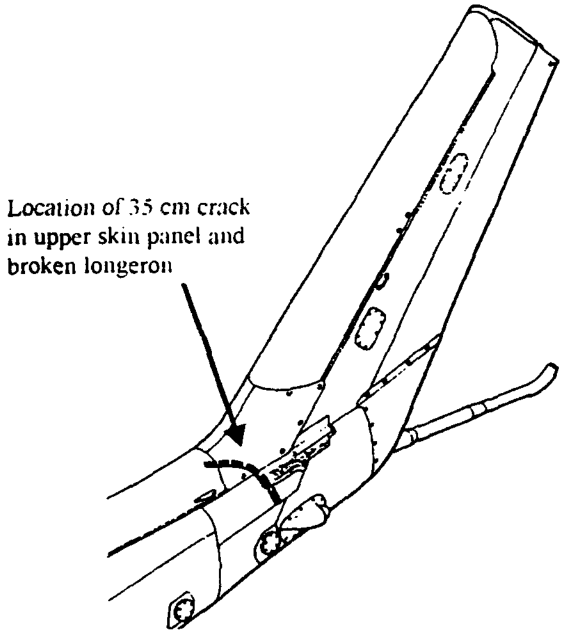
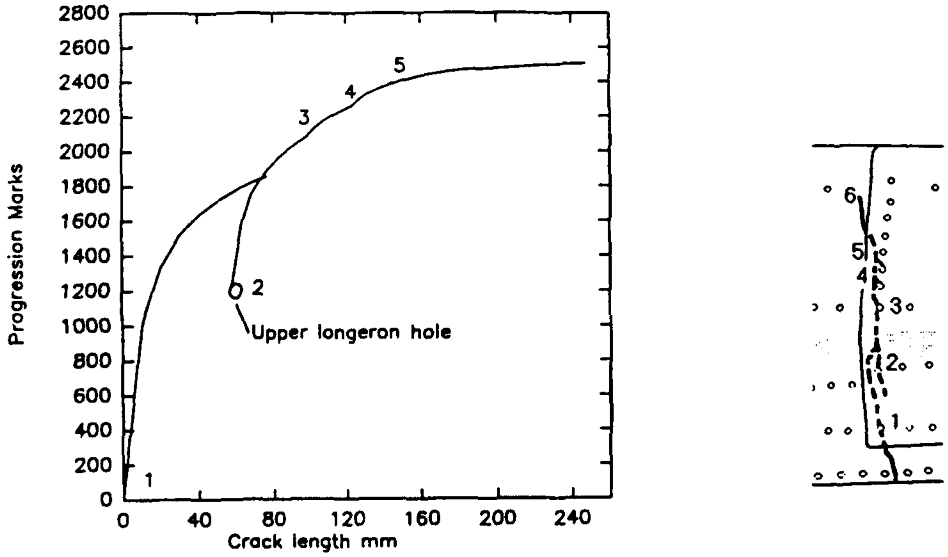
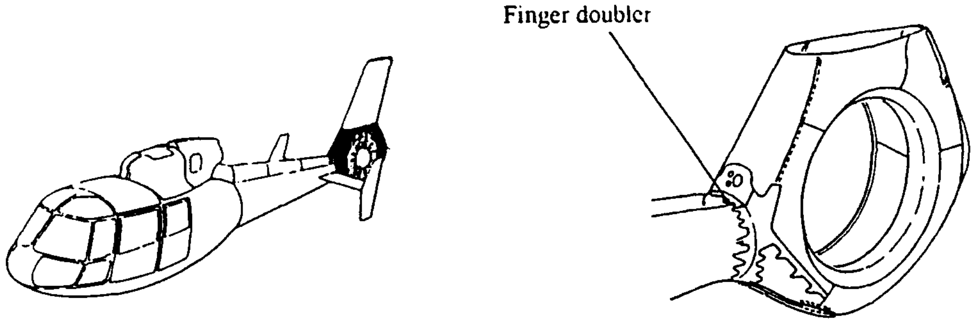


FIG. 3 BELL 212 CRACK LOCATION



Crack growth rate of the skin crack from the lower longeron hole to the end of the crack

FIG. 4 BELL 212 CRACK GROWTH CURVE



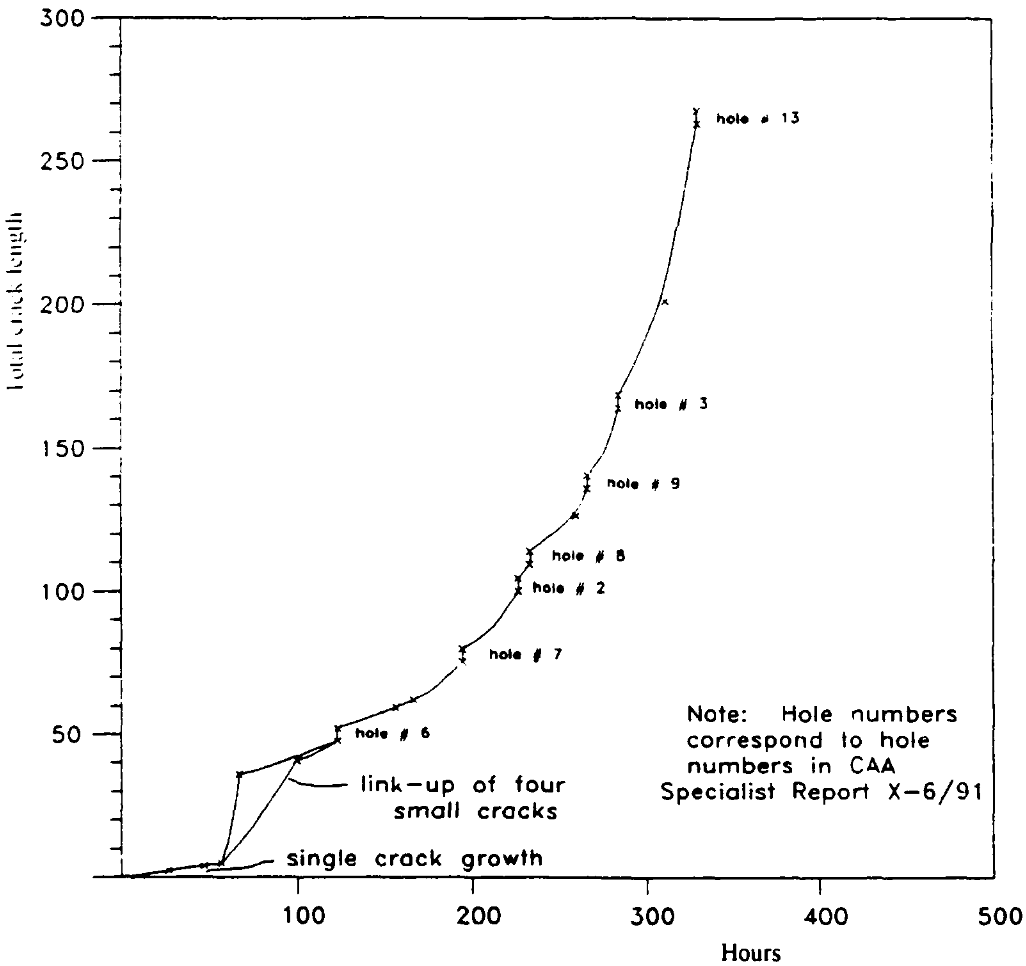


FIG. 5 DAUPHIN CRACK GROWTH CURVE

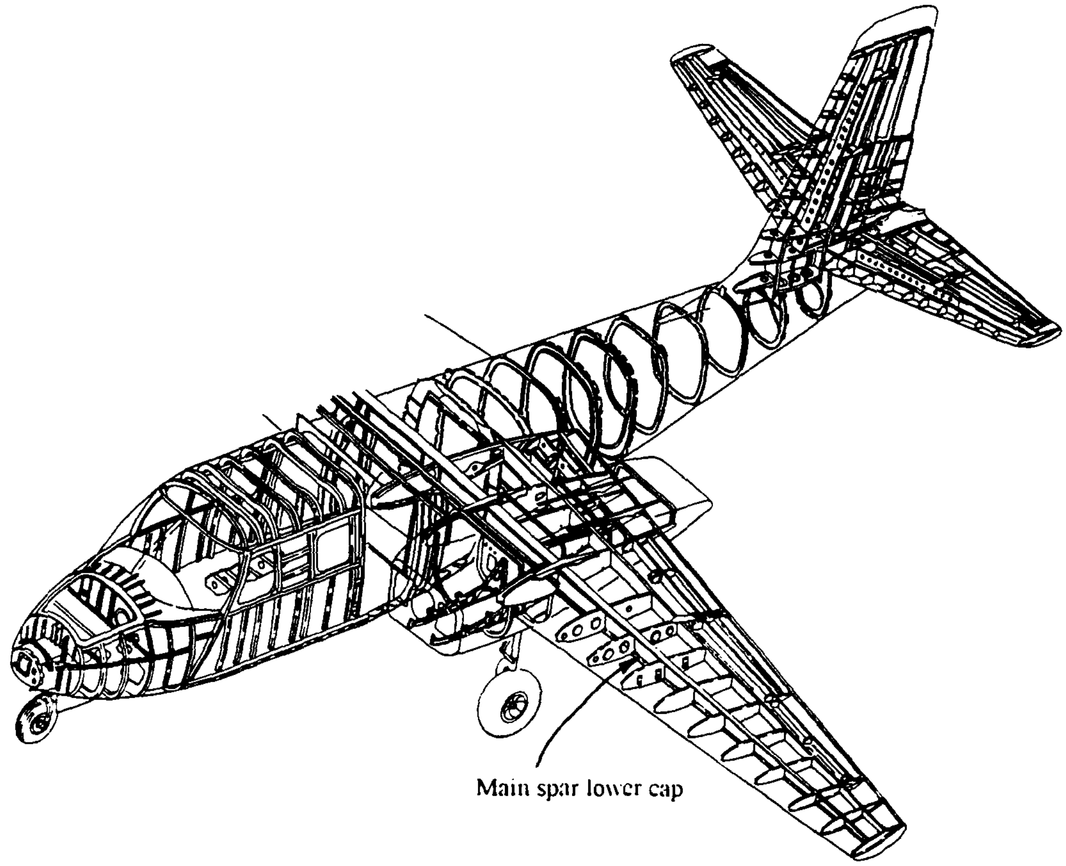
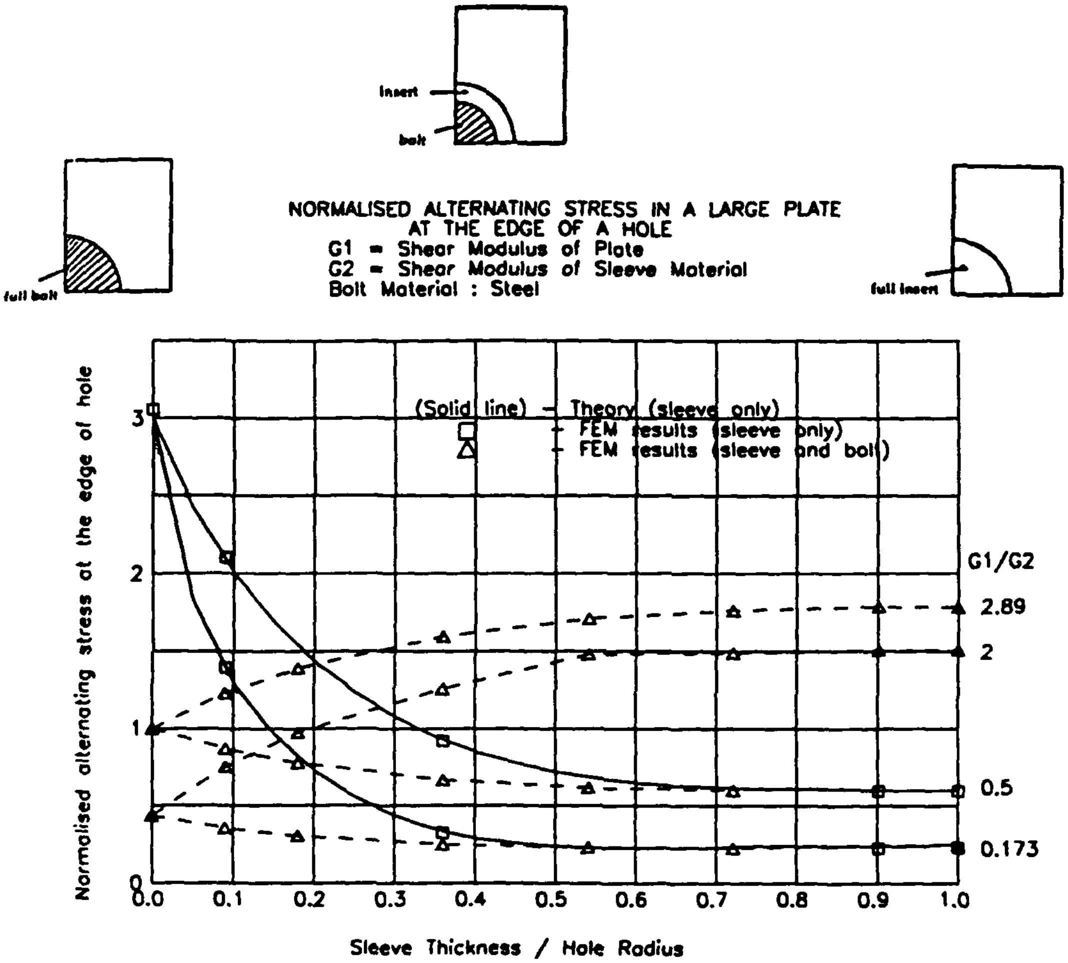


FIG. 6 AERO COMMANDER CORROSION



Note: for sleeve and bolt cases, bolt radius + sleeve thickness = hole radius

FIG. 7 FINITE ELEMENT RESULTS FOR ALTERNATING STRESSES USING SLEEVE/BOLT COMBINATIONS FOR HIGH AND LOW MODULUS SLEEVES AND PLATES

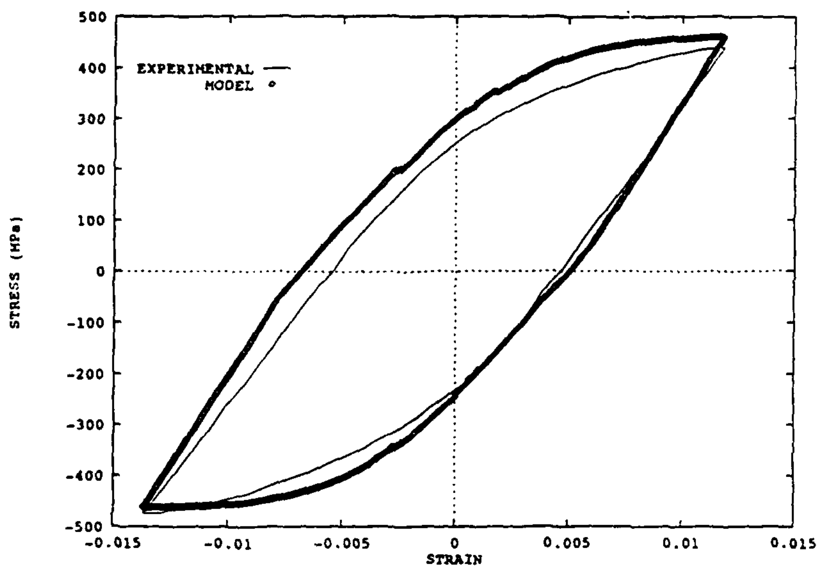


FIG. 8 UNIAXIAL RESPONSE OF AL 7050

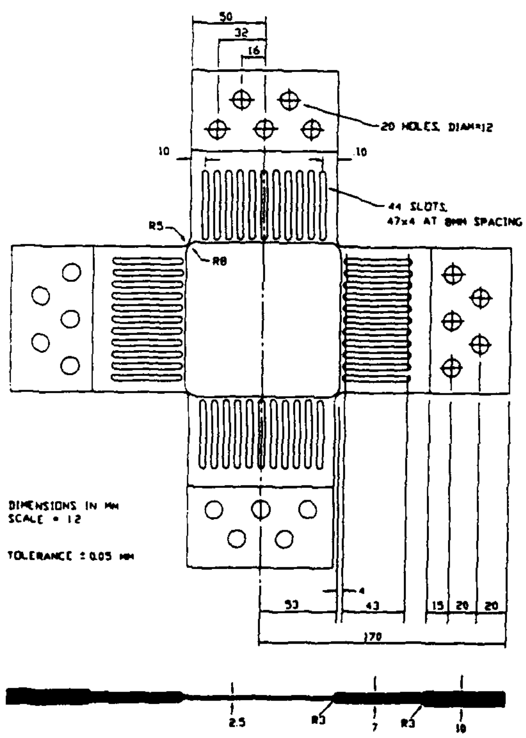


FIG. 9 BIAXIAL TEST SPECIMEN

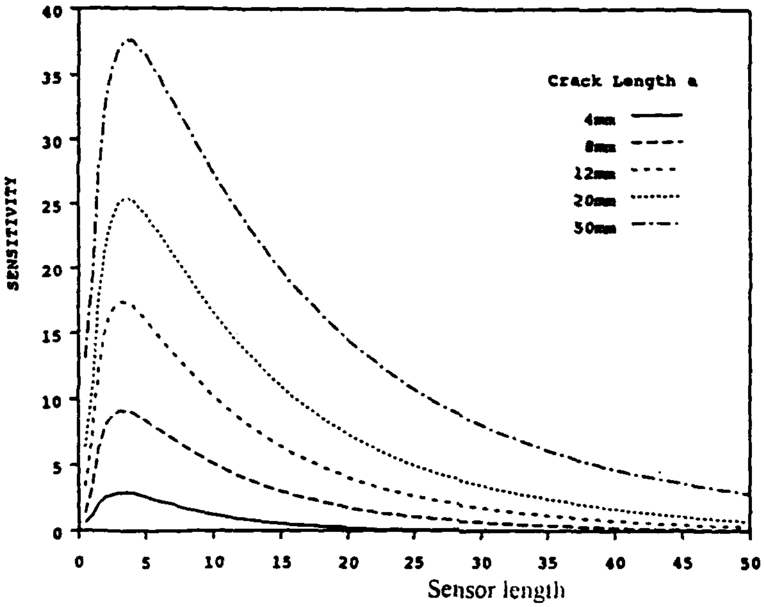


FIG. 10 SENSITIVITY OF PIEZO SENSOR FOR VARIOUS CRACK LENGTHS

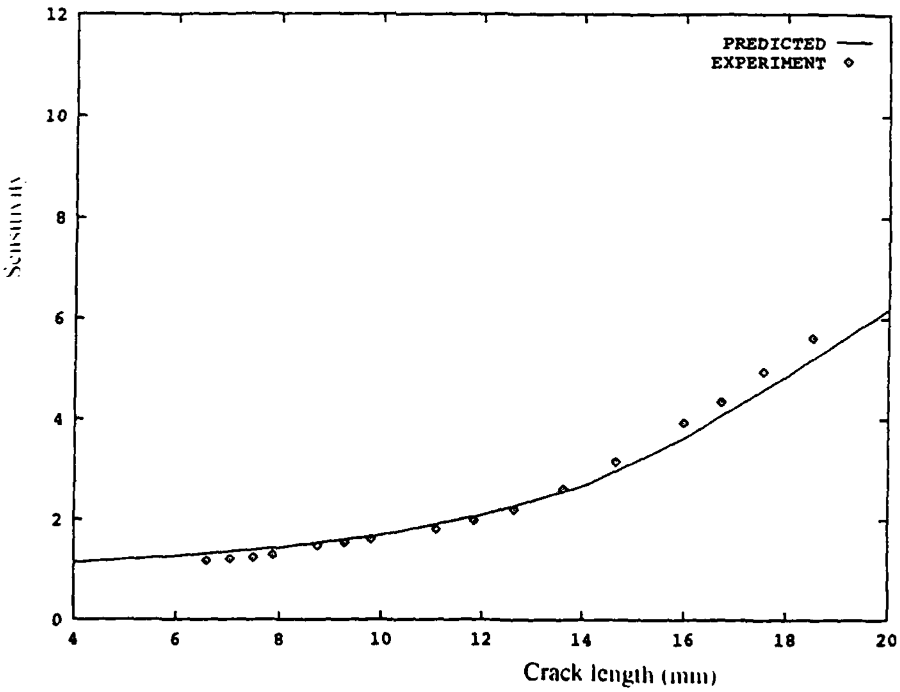


FIG. 11 THEORETICAL AND EXPERIMENTAL OUTPUT FROM PIEZO SENSOR WITH INCREASING CRACK LENGTH

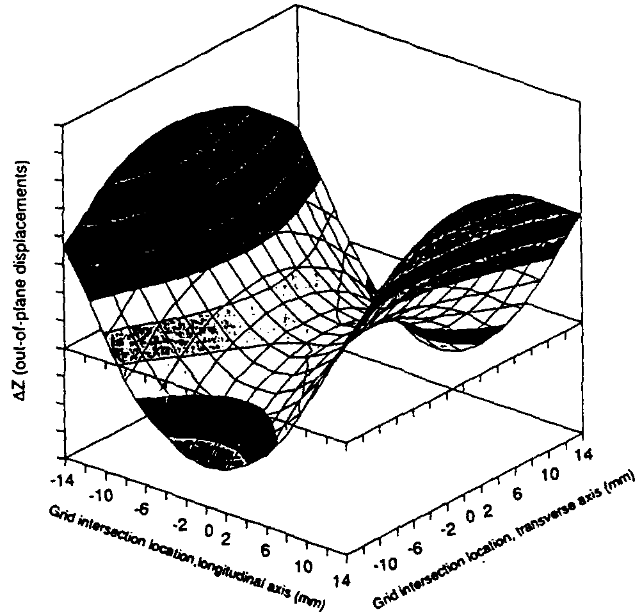


FIG. 12 OUT-OF-PLANE DISPLACEMENTS OF COMPRESSION SURFACE OF BEND SPECIMEN (MEASURED THEN SMOOTHED)

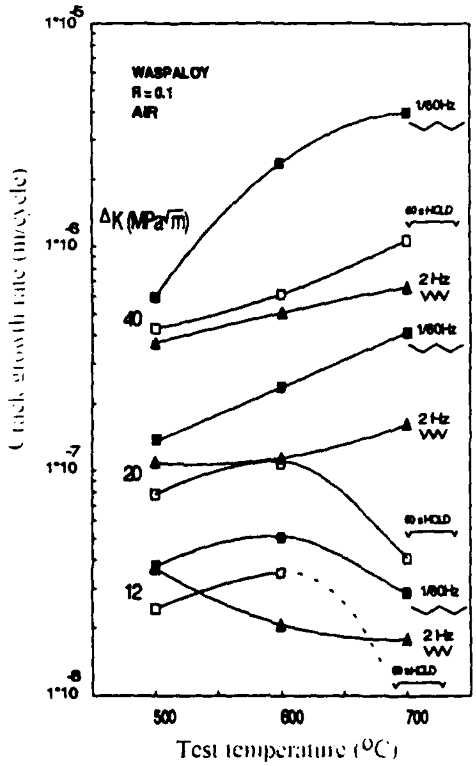


FIG. 13 VARIATION IN CRACK GROWTH RATE AT VARIOUS TEST TEMPERATURES FOR A VARIETY OF CYCLIC FREQUENCIES AND HOLD TIMES

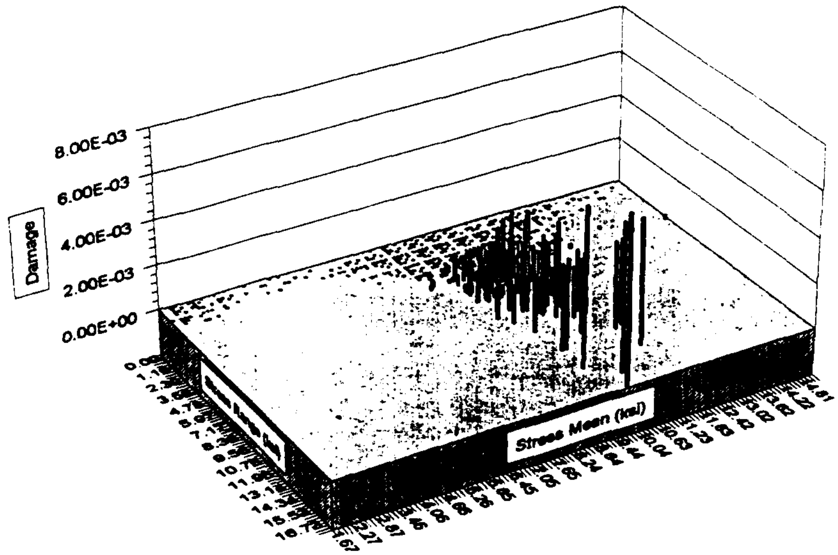


FIG. 14 WING FATIGUE DAMAGE AS A FUNCTION OF STRESS RANGE AND MEAN

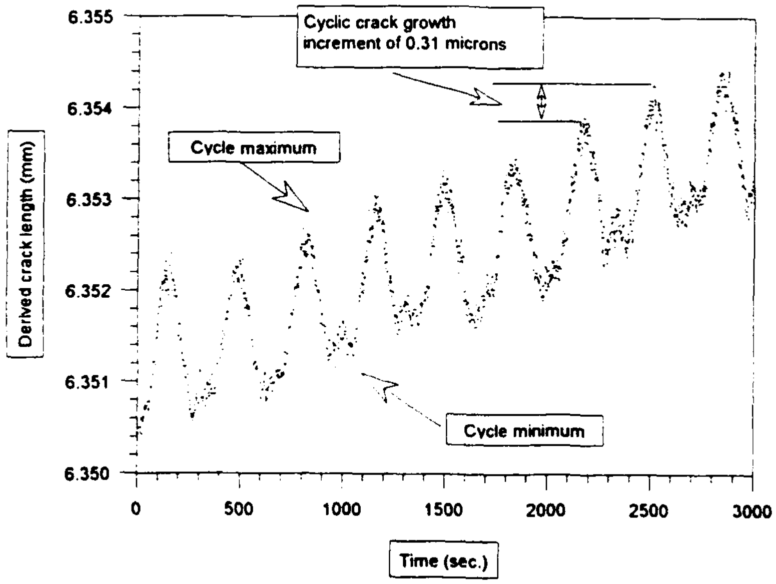


FIG. 15 INCREMENTAL CRACK GROWTH EFFECTS DISCERNIBLE IN ELECTRICALLY MEASURED VARIATIONS IN CRACK TIP OPENING BEHAVIOUR OF A CRACKED SENB SPECIMEN UNDERGOING LOW FREQUENCY CYCLIC LOADING

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16. ABSTRACT

This document was prepared for presentation to the 23rd Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Stockholm, Sweden on June 7 and 8, 1993.

A review is given of the aircraft fatigue research and associated activities which form part of the programmes of the Aeronautical Research Laboratory, Universities, the Civil Aviation Authority, the Australian aircraft industry and the Defence Scientific Establishment, New Zealand. The review summarises fatigue-related research programmes as well as fatigue investigations on specific military and civil aircraft.

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